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TECHNOLOGY UTILIZATION

COMPOSITE MATERIALS

A COMPILATION

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Foreword

The National Aeronautics and Space Administration has established a Technology Utilization Program for the dissemination of information on technological developments which have potential utility outside the aerospace community. By encouraging multiple application of the results of its research and development, NASA earns for the public an increased return on the investment in aerospace research and development programs.

Compilations are now published in nine broad subject groups:

SP-5971: Electronics - Components and Circuitry
SP-5972: Electronics Systems
SP-5973: Physical Sciences
SP-5974: Materials
SP-5975: Life Sciences
SP-5976: Mechanics
SP-5977: Machinery
SP-5978: Fabrication Technology
SP-5979: Mathematics and Information Sciences

When the subject matter of a particular Compilation is more narrowly defined, its title describes the subject matter more specifically. Successive Compilations in each broad category above are identified by an issue number in parentheses: e.g., the (03) in SP-5972 (03).

In this Compilation articles are presented on both metallic and nonmetallic composite materials. Three sections describe their properties, handling, and application.

Additional technical information on items in this Compilation can be requested by circling the appropriate number on the Reader Service Card included in this Compilation.

The latest patent information available at the final preparation of this Compilation is presented on page 23. For those innovations on which NASA has decided not to apply for a patent, a Patent Statement is not included. Potential users of items described herein should consult the cognizant organization for updated patent information.

We appreciate comment by readers and welcome hearing about the relevance and utility of the information in this Compilation.

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Contents

Section 1. Materials and Material Properties

BORON-ALUMINUM COMPOSITE STRUCTURES

Design, analysis and fabrication techniques have been developed for boron-aluminum composite-structure technology. They were compared with those of conventional metal-structure technology to evaluate relative performance. In the developmental program, efforts were concentrated toward developing the capacity to design and analyze boron-aluminum structures and to fabricate the structures in a production shop. These efforts were based on a eutectic bonding process for joining monolayer boron-aluminum foils into laminate composite parts.

The program involved the formulation of a material specification for the boron fibers, specifically for use in aluminum-matrix composites and for the monolayer boron-aluminum foil sheets used as the starting material in the production process. Specifications were initiated also for the eutectic bonding methods. Investigations were made of the various tools, machines, and techniques that were needed to perform the drilling, countersinking, routing, machining, cutting, and trimming required to fabricate the test panels. These investigations were performed to transfer information, already developed in the laboratories, to the production shop. Many tests were performed to determine the strength, elasticity, and local instability, as well as joint properties, of the boron-aluminum composite fabricated in the production shop.

Additional studies were conducted to select the configuration and to proportion the elements of the test panel. These included investigation of stringer shape, thickness, and spacing; skin thickness and frame spacing; and configuration and material distribution. The results of the studies included tradeoffs among stringer shape and fastener requirements, frame design to achieve shear transfer from the skin, tension clip effects on the stringer, and requirements for joggles and in joints. These data were used in the design of a 48-inch by 61.5-inch (1.21-m by 1.56-m) test panel. Tests then were conducted on the panel in two sequences, the first involving loading to 30 percent of design ultimate load at room temperature. The sequence was compression, unload, pure shear, unload, and combined compression and shear. The second sequence involved loading to failure in a stepped sequence at 500° F (260° C).

The results of these studies show a high weight-savings potential provided by the strength of boron-aluminum structures. The developed composite structures are primarily for application on Space Shuttle vehicles; but could be advantageous in other areas, particularly the aircraft industry.

Source: R. E. Jackson of
McDonnell Douglas Corp.
under contract to
Marshall Space Flight Center
(MFS-21571)

Circle 1 on Reader Service Card.

FATIGUE OF BORON-ALUMINUM COMPOSITES BONDS AND JOINTS

Composites hold considerable promise as lightweight, high modulus, corrosion resistant, structural materials. There are, however, several technical problems to be overcome before composites can be widely used. A study of one such problem, the bonding and joining of composites, is discussed in detail in a recent study.

The effects of boron filament diameter on bonds and joints are examined for boron-aluminum composites. The data developed were not previously available and will be of interest to many industries considering the development of composite technology.

The data include static strength, fatigue, and dynamic moduli of elasticity; Manson-Coffin analyses and metallurgical and fracture surface evaluation were also performed. The joining methods examined are diffusion bonding, aluminum dip brazing, low-temperature brazing, resistance welding, riveting, and mechanical fastening.

The intent of the study has been to provide a preliminary view of fatigue of B/Al joints. In addition to the substantial quantity of data accumulated, several conclusions concerning B/Al composites were drawn.

a. The static and fatigue strength of B/Al, unidirectionally reinforced with 0.14 mm (5.6 mil) boron, is superior to that of B/Al reinforced with 0.10 mm (4.0 mil) boron.
b. The static and fatigue strength of some joints in B/Al reinforced with 0.10 mm (4.0 mil) B is superior to those joints reinforced with 0.14 mm (5.6 mil) B.
c. The fatigue strength of Hi-Lock fastener joints is superior to that of resistance weld or rivet joints.
d. The fatigue strength of low-temperature braze joints is inferior to that of dip braze or diffusion bond joints.

Source: M. S. Hersh of
General Dynamics Corp.
under contract to
Marshall Space Flight Center
(MFS-22325)

Circle 2 on Reader Service Card.

GRAPHITE-REINFORCED ALUMINUM COMPOSITE

Metallic composites are a relatively new family of structural materials that combine the desirable properties of high strength, high-tensile modulus, and low density. At the same time, they are usable over a wider temperature range than ordinary aluminum or magnesium alloys. The composites are generally formed by using a matrix of lightweight metal (aluminum or magnesium) and incorporating oriented high-strength reinforcing fibers of a high-modulus low-density refractory material such as boron or graphite.

A new aluminum composite, reinforced with nickel-plated graphite fibers, is prepared by applying an electroless nickel coating to graphite-fiber yarn, alining the yarn between aluminum sheets in a stacked array, and heating the array to obtain diffusion bonding. This composite has several desirable features. For instance, the graphite fibers remain intact throughout the process, and undesirable aluminum/carbon reactions at the fiber surface are prevented by the nickel coating. Furthermore, high-strength graphite fiber is much less expensive than some other commonly-used reinforcing materials; it has a greater potential for future reductions in manufacturing costs; and, unlike other fibrous substances (particularly boron), it yields a composite material that can be bent or otherwise formed without breaking the fibers.

Source: F. P. Lalacona
Marshall Space Flight Center
(MFS-21077)

Circle 3 on Reader Service Card.

TITANIUM-REINFORCED BORON-POLYIMIDE COMPOSITE

High-modulus reinforced composites are being increasingly used due to their higher strength-to-weight and stiffness-to-weight ratios. The application of optimized, oriented, filamentary composite structures in aerospace applications has been limited by low stiffness, bearing characteristics and, in some instances, temperature deterioration. The low stiffness (composite modulus) has been overcome by the development of boron and high-modulus graphitic filaments. The elevated temperature characteristics have been improved by such polymers as polyimides, polybenzimidazoles, and polybenzothiazoles. These resin systems have not, however, been completely characterized for their processing characteristics, particularly as they apply to production-type tooling. Bearing characteristics and concentrated load transfer through hard points still remain problem areas where minimum weight is required.

One good approach to reducing the weight of hard points is to incorporate lightweight metal shims to augment the strength of the composite. To minimize strains at the metal-composite interface, the component must have compatible linear coefficients of thermal expansion. A boron composite with any resin system is reasonably compatible with stainless steels, but is more compatible with titanium sheet materials. Graphite composites impose problems in the incorporation of metal shims, since they have a slightly negative coefficient of expansion in the longitudinal direction.

The intended use of any material imposes considerations on how it will be processed. One application of interest considered in the development effort, but not to be done under this program, was the fabrication of a conical section 9.75 m (32 ft) in diameter at the base, tapering to 8.65 m (28 ft) in approximately 2 m (6 ft). The size of such a section obviously precludes the press molding of the component in one piece. It also severely limits the facilities for autoclave or hydroclave molding. Thus, the processing technique established for concentrated effort was conventional vacuum-bag molding.

This program involved the development of the process technique for boron-polyimide prepreg; the layup and curing procedures for the prepregs when processed under vacuum-bag pressure; and the development and evaluation of titanium hard points, for the smooth transition of loads from the titanium attach points into the boron-reinforced body of the structure. After data generation, this information was applied to the modification of a box beam design which had been fabricated previously of boron/epoxy. After verification of the design, the component was fabricated and tested structurally at the maximum temperature capability of the system.

Source: G. A. Clark of
Rockwell International Corp.
under contract to
Marshall Space Flight Center
(MFS-21916)

Circle 4 on Reader Service Card.

DEVELOPMENT OF BERYLLIUM HONEYCOMB-SANDWICH COMPOSITE FOR STRUCTURAL AND OTHER RELATED APPLICATIONS

A detailed report is available that deals with the feasibility of replacing steel, aluminum, or fiberglass honeycomb panels with a beryllium composite. This composite should be particularly useful where rigid lightweight structures are needed.

In the report, face and core materials, cutting, forming, bonding, and brazing are described. The extensive data developed are useful, not only in the development of these panels but also in the development of braze-alloy composites in general.

Radiographic inspection of three 0.9-m^2 (10-ft^2) panels showed a 97 percent braze with excellent edgewise compression, flatwise tension/compression, and block-shear values up to 590 K. The panels also had good vibrational loading characteristics with desirable damping.

Source: J. W. Vogan and L. A. Grant of
International Harvester Co.
under contract to
Marshall Space Flight Center
(MFS-22633)

Circle 5 on Reader Service Card.

NICKEL-PLATED POLYIMIDE FIBERGLASS

Polyimides have been used to reinforce glass laminates in several special applications. This type of composite material combines the very high theoretical strength of glass with the stability and flexibility of polyimides. The composite material, a type of fiberglass, has greater thermal stability and resistance to flame than conventional fiberglass.

A report on the development of polyimide fiberglass, including layup dies and curing, has been prepared. Two phases of the development effort are of particular interest: (1) the development of polyimide tools and (2) a nickel (copper or silver) plating process, which produces a nonporous metallic laminate with increased strength and a mirrorlike surface finish.

The metal-plated laminate is most versatile. It is stronger without much added thickness, it is light-reflective, and it conducts electricity to about the same degree as the plating metal.

Source: N. R. Campbell and J. R. Mix of
Rockwell International Corp.
under contract to
Johnson Space Center
(MSC-19188)

Circle 6 on Reader Service Card.

GRAPHITE AND BORON-REINFORCED COMPOSITE MATERIALS DATA SUMMARY

A data summary file has been assembled consisting of currently-available comprehensive information concerning graphite and boron-reinforced composite materials. The data summary is not intended to be a detailed design guide, but rather a collection of information on typical processing techniques, mechanical properties, and physical properties of the advanced composite materials, which are being considered for structural applications on advanced space vehicles.

Information in the form of tables and graphs is provided covering shear modulus, shear strength, flexure modulus, flexure strength, stress-strain data, compressive modulus, compressive strength, interlaminar shear strength, linear thermal expansion, longitudinal and transverse tension strength, specific heat, thermogravimetric and differential thermal analysis, time-temperature-viscosity relationships, electrical resistivity, and hoop data. The effects of room temperature and elevated temperature aging or post cure are shown. Available data on many commercially-available composite materials are included.

These composite materials were developed primarily to fill aerospace industry needs, providing good strength at elevated temperatures with high stiffness and low weight. However, they will have many uses in other fields where their unique properties can be utilized, such as prosthetics and rehabilitation equipment and marine masts, booms, and spars.

Source: General Dynamics
under contract to
Marshall Space Flight Center
(MFS-21691)

Circle 7 on Reader Service Card.

HIGH-STRENGTH LARGE-DIAMETER CARBON-BASE FIBERS

Large-diameter [0.005 to 0.025 cm (0.002 to 0.01 in.)] high-strength carbon-base monofilaments have been prepared by the pyrolytic vapor-phase deposition of carbon (or codeposition of carbon and boron) onto a carbon-fiber substrate. The new material is primarily applicable as a reinforcement for metal-matrix composites.

Although conventional carbon yarns, consisting of small diameter [0.0008 cm (0.0003 in.)] multifibers, can effectively reinforce resin-type matrix composites, such yarns have very limited use in metal-matrix composites because they are degraded by chemical reaction with the matrix material. When the conventional fibers are coated to reduce this degradation, other problems arise.

The diameters of the new fibers are large enough to permit the use of a conventional lay-up type of fabrication and the application of sufficiently-thick protective coatings, without reducing the effective fiber content. The monofilament form allows the use of state-of-the-art fabrication techniques for metal-matrix composite reinforcements such as boron and silicon carbide. The development effort for these large-diameter, carbon-base monofilaments has been primarily in the 0.008 to 0.16 cm (0.0003 to 0.006 in.) diameter range.

The process is conducted by passing a carbon filament substrate [0.0005 cm (0.002 in.) diameter] through a chemical vapor-deposition chamber at nominal velocities of 15 to 46 cm/min (0.5 to 1.5 ft/min) and a temperature of at least 1,073 K (1,073 to 2,473 K range). The atmosphere within the chamber contains a hydrocarbon gas such as the alkane or alkene series, with or without additions of a boron-containing gas. If only a hydrocarbon gas is used, monofilaments having an ultimate tensile strength (UTS) of approximately 1.03×10^9 N/m^2 (150,000 psi) are easily produced from substrate filaments with an initial UTS of approximately 0.7×10^9 N/m^2 (100,000 psi). If the atmosphere also includes a boron-containing gas, fibers having a UTS of approximately 3.4×10^9 N/m^2 (500,000 psi) are possible. With such high tensile-strength and modulus-of-elasticity values of 241×10^9 N/m^2 (35,000,000 psi), these large-diameter, carbon-base fibers represent a considerable improvement over conventional carbon-base yarns.

The carbon-base monofilaments are also better than conventional boron filaments on a specific-strength basis. Carbon-base monofilaments retain their desirable properties at elevated temperatures, whereas boron filaments show a catastrophic reduction in strength and in other properties at temperatures above 1,030 K (1,395° F).

Metal-matrix composites containing these new large diameter, carbon-base monofilaments offer the advantages of superior strength at elevated temperatures and very low density [0.19 kg/m^3 (0.012 lb/ft^3)]. They therefore should prove useful in high-temperature equipment where component weight must be minimized.

The following documentation may be obtained at cost from:

National Technical Information Service
Springfield, Virginia 22151

Reference: NASA-CR-72770 (N71-17328/LK), Development of Manufacturing Process for Large-Diameter Carbon-Base Monofilaments by Chemical Vapor Deposition

Source: R. L. Hough of
Hough Laboratory
under contract to
Lewis Research Center
(LEW-11167)

FIBER COMPOSITE MATERIALS: A SURVEY OF FIBER/MATRIX INTERFACE MECHANICS

A state-of-the-art survey has been made and a summary report published on the mechanics of load transfer at the fiber/matrix interface of fiber composite materials.

It is well known that the structural integrity of fiber composites derives from the fiber/matrix interface, the bond between fiber and matrix and the region immediately adjacent. The significance of the interface is indicated by the realization that a 16.39 cubic centimeter (1 cubic inch) volume of a 50-volume-percent composite with 0.000762 centimeter (0.0003 inch) diameter fibers contains about 41,925 square centimeters (6500 square inches) of interface area. At least three types of bonding are believed to exist at the interface: chemical, electrical, and mechanical. Irrespective of the nature of the bond, however, load transfer is primarily a mechanistic process.

This survey report discusses the mechanism of load transfer from matrix to fiber through the interface and the effects of the interface on composite structural integrity. Specifically addressed are the role of the interfacial bond in composite strength, the dependence of fracture surface on the interface bond strength, methods for measuring and predicting stress at the interface, the microresidual stress and load condition effects on the interface bond, and the effects of voids and fiber breaks on the interface bond. The possibility of designing composites with specified bond strengths is examined. General trends and significant points are illustrated graphically.

The report discusses theoretical considerations supplemented with experimental data. Information drawn from the published literature is supplemented by recent results obtained by the author and his associates to project some current trends and thinking. The results examined and summarized lead to the following conclusions:

1. The better the interface bond, the higher the static composite strength.
2. High interface bond strength results in brittle and notch-sensitive composites.
3. Typical tests for interface bond strength include: matrix Poisson effect, fiber pullout, fiber push-through, short-beam-horizontal shear, transverse tensile, dynamic modulus, and photoelasticity.
4. Appropriate methods of analysis include: mechanics of materials, classical elasticity, and finite element.
5. Microresidual stresses and longitudinal loads along the fiber direction set up forces which tend to break the bond.
6. The presence of voids at the interface weakens the bond.
7. Elevated-temperature environments degrade the interface bond.
8. When a fiber breaks within the composite, the energy released tends to debond the interface.
9. The matrix effects on the bond are indicated by a debonding parameter. This debonding parameter can be used to design the interface bond for specific composite applications.

This report contains a comprehensive discussion of work reported through mid-1971 on all aspects of the fiber/matrix load transfer mechanics and includes 60 pertinent references. It is a valuable source of integrated information for researchers and practitioners in the fiber composite community and should be particularly useful to fiber, matrix, and composite producers, experimenters, analysts, designers, and basic researchers. It also provides an abundance of material for courses of study in composite mechanics.

The following documentation may be obtained at cost from:

National Technical Information Service
Springfield, Virginia 22151

Reference: NASA TN-D-6588 (N72-16885/LK), Mechanics of Load Transfer at the Fiber/Matrix Interface

Source: Christos C. Chamis
Lewis Research Center
(LEW-11924)

MOLDABLE, RESILIENT, AND NONABRADING POLYIMIDE FOAM

A moldable, resilient, and nonabrading foam may be prepared from a commercially-available polyimide powder. As normally prepared, the powder yields a coarse inhomogeneous nonisotropic nonmoldable foam with a density of 0.6 lb/ft^3 (9.6 kg/m^3). By using a special process, a new homogeneous foam can be made. It has a high degree of resiliency, a nonabrading skin surface, and an open-pore structure. It can be molded easily and may be prepared in densities ranging from 2 to 12 lb/ft^3 (32 to 192 kg/m^3).

The new foam may be prepared as follows:

1. The low-density polyimide powder is heated in an oven to 345° F (175° C) for a short period of time (30 minutes), until a red friable foam has been formed. This red "B" stage foam is then further cured for 30 minutes, at 570° F (300° C), to form a strong tough buff-colored fully cured foam.
2. The low-density foam is shredded in a blender (or equivalent device), to give a lightweight high-bulk material with apparent density of 0.5 lb/ft^3 (8 kg/m^3 or less. The cured foam can be shredded and not ground, to leave a considerable amount of the initial foam structure intact. Individual shredded particles can be as large as 1/4 in. (0.06 cm) in diameter, without adversely affecting the molding process. Less than 50 percent by weight of the shredded product should pass through a 20-mesh screen.
3. This shredded foam then is placed in a mold and compressed to the volume calculated to give the desired density. It is heated between 570° and 545° F (300° and 285° C) for 4 to 8 hours. Uniform heating of the mold is required to produce a well-fused product. However, higher molding temperatures tend to cause the foam to stick to the metal surfaces of the mold.
4. After heating for the required amount of time, the mold is cooled, and the fused molded product removed.
5. If desired, the foam can be impregnated with phosphorus-containing compounds to improve resistance to burning in high-oxygen environments.
6. Cut edges of the foam can be sealed with a nonabrading skin, by hot-ironing the surface while it is covered by aluminum foil.

The following documentation may be obtained at cost from:

National Technical Information Service
Springfield, Virginia 22151
Reference: NASA CR-115692 (N75-76139)

Source: G. L. Ball III, I. O. Salyer,
and G. R. Wilson of
Monsanto Corp.
under contract to
Johnson Space Center
(MSC-14315, 14152)

NONFLAMMABLE STRUCTURAL COMPOSITES

A new structural composite is nonflammable, even in very thin sections. It is made from a resin/fiber combination that can be easily laminated; for example, typical production requires a pressure of 10.3×10^4 N/m^2 (15 psi), at 533 K (500° F) for 2 hours.

A glass fabric is impregnated with polyquinoxaline resin by a conventional method such as solution coating. The prepreg is stacked in the desired thickness and form and placed in a press or an autoclave. The result is a dense, structurally sound composite.

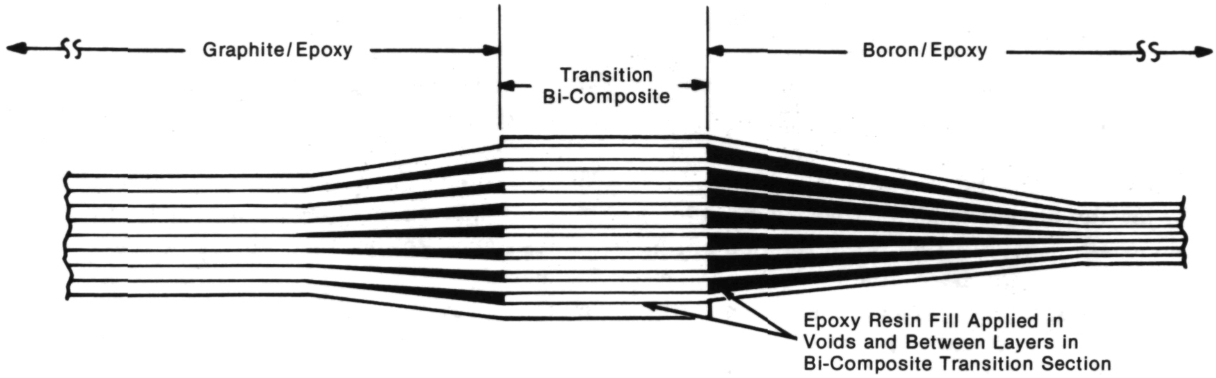
There is some evidence that graphite fiber may be used in place of glass, to reduce the weight and increase stiffness.

Source: C. L. Segal and G. Fox of
Whittacker Corp.
under contract to
Johnson Space Center
(MSC-13496)

Circle 8 on Reader Service Card.

Section 2. Techniques for Fabrication and Use

A NEW CONCEPT FOR JOINING DISSIMILAR COMPOSITES



Bi-Composite Joint

Various parts of a structure fabricated from composite materials may have different uses or be subject to different stresses. If the structure is made from a combination of different composites, the best composite, chosen for its properties and for economy, may be used for each part of the structure. However, there has been no suitable way of joining unlike composites without introducing weaknesses or increasing the weight of the structure.

A new method of joining different laminated composites without mechanical fasteners has been proposed. Structures of more than one kind of composite may be formed by interleaving the plies of one composite with the plies of another. In this way, the properties of each composite may be tailored to the requirements of a given structure.

This bi-composite joint serves as an interface between two dissimilar materials. Composites are normally fabricated by means of a tape layup in which plies of the materials are stacked in a definite orientation to provide the desired mechanical properties. A bi-composite joint is made by interleaving the plies of one composite with the plies of another. The method is illustrated in the figure.

The interleaving forms a transition area between the composites. Voids in this area are filled in with epoxy resin to form a strong, smooth transition between the two materials.

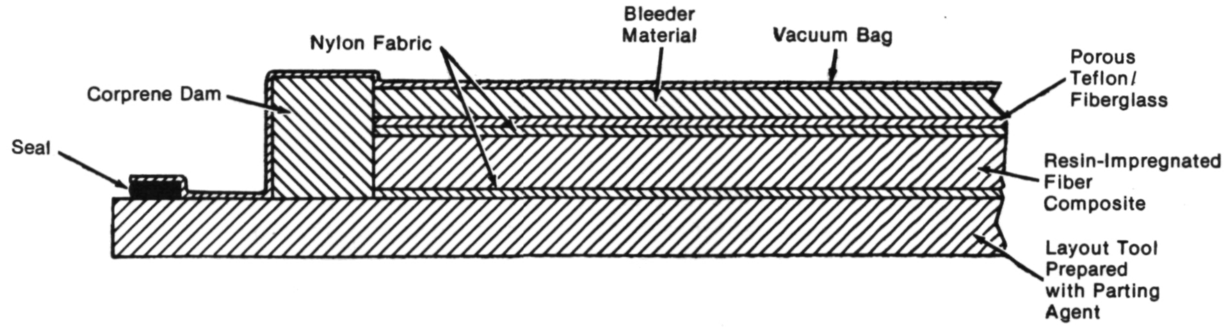
There are several advantages to this method:

1. Lower cost laminates can be used in parts of the structure where more expensive composites are not necessary.
2. More flexible fibers can be used where a structure has intricate contours.
3. Properties can be tailored to the structure. For instance, a low thermal conductivity composite may be used where heat insulation is required, and a high thermal conducting fiber where heat conduction is desired.
4. Virtually any combination of composite materials can be joined.

Source: K. C. Dullea and
J. A. Evangelista of
Rockwell International Corp.
under contract to
Marshall Space Flight Center
(MFS-24307)

Circle 9 on Reader Service Card.

BONDABLE COMPOSITE SURFACES



Cross Section of a Typical Vacuum-Bag Autoclave-Cure Setup as Described in the Text

An industry using fiberglass, boron/resin matrices, or graphite laminates would profit by being able to eliminate abrasive surface cleaning, usually necessary to achieve a secondary bondable surface.

This can be accomplished using a new method: the application of a tear ply of thin nylon fabric during the molding operation. After molding, the fabric simply is peeled from the laminate. This removes surface-resin gloss, leaving an ideal surface for metallic or nonmetallic components.

The specific steps for the fabrication of the composite are as follows:

1. Mold preparation:
   Apply a parting agent to the mold surfaces, and bake it on for 30 minutes at 445 K (350° F).
2. Laminate layup and cure:
   a. Apply a layer of nylon fabric to the mold areas that correspond to the part to be bonded.
   b. Stack the resin-impregnated-fiber composite over the prepared mold and the nylon-fabric surface.
   c. Apply additional nylon-fabric tear-ply material (a single layer) to the top surface of the stacked resin-fiber prepreg, where adhesive bonding operations will be required.
   d. Apply one layer of Teflon-coated porous-fiberglass separator film over the nylon tear ply and stacked prepreg.
   e. Assemble bleeder materials, as required, for removing excess resin from the prepreg.
   f. Mold the prepared stacked laminate in an autoclave, an oven, or a press, per the required prepreg curing schedule. (A typical autoclave vacuum-bag molding operation for graphite/epoxy material is presented in the figure.)
3. Removal of bleeder materials and nylon-fabric tear-ply layer:
   a. Remove the vacuum bag and the bleeder material, simply by peeling loose the porous Teflon-coated fiberglass and the bleeder materials.
   b. Remove the nylon-fabric tear ply, by securing an edge and peeling it back, parallel to the laminate surface.
   c. Remove the nylon-fabric tear-ply material next to the mold surface.
4. Autoclave-cure process:
   a. Apply a vacuum of 63.5 cm (25 in.) of mercury.
   b. Slowly increase the temperature from room temperature to 405 K (270° F), and hold it there for 30 minutes.
   c. Apply 72×10^4 to 79×10^4 N/m^2 (90 to 100 psig) to the autoclave.
   d. Slowly raise the temperature to 445 K (350° F); cure for 2 hours.
   e. Cool the composite below 340 K (150° F) before releasing the autoclave pressure.

Source: J. S. Jones of
Rockwell International Corp.
under contract to
Marshall Space Flight Center
(MFS-24469)

Circle 10 on Reader Service Card.

CRYOGENIC DEBONDING

Because force is needed to remove a bonded insert from a fiberglass phenolic laminate, the parent material may crack. With a new method, the insert can be removed rapidly and without damage to the laminate. The procedure is as follows:

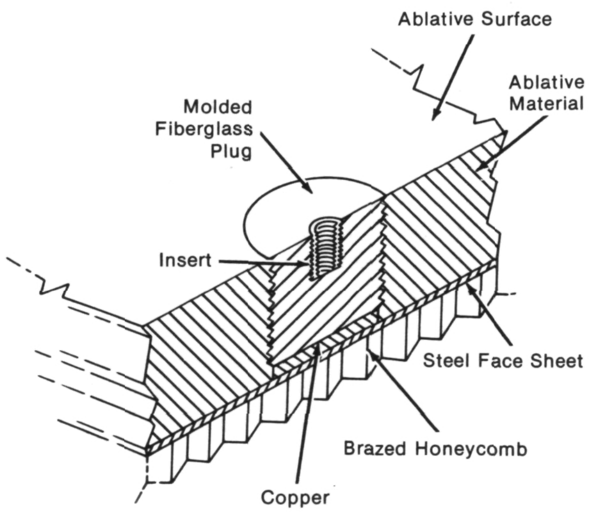
a. Attach a flexible metal hose to a liquid nitrogen source. Terminate the hose with an orifice, nozzle, or perforated tube, suitably designed to fit into the cylindrical bore for a maximum surface wetting rate.
b. Pressurize the liquid nitrogen source sufficiently to produce the desired flow rate of nitrogen vapor. Chill for 60 ±10 seconds.
c. Tap in a conventional easy-out tool, and apply enough torque so that the insert withdraws in a slow steady motion.
d. After removing the insert, dry the residual moisture on the parent phenolic material with a heat gun, a blanket, or an oven.
e. Remove any residual adhesive from the threaded recess in the phenolic, and reinstall a new threaded insert.

This technique requires no elaborate equipment; it can be used in any shop that has a liquid nitrogen source. It could result in immediate savings to manufacturers of boats, aircraft, or electronic-circuit boards.

Source: F. Mandaro and G. J. Cizek of
Rockwell International Corp.
under contract to
Johnson Space Center
(MSC-15444)

No further documentation is available.

PLUG FOR JOINING ABLATIVE MATERIAL



Plug for Joining Ablative Material

A fiberglass plug can form a strong bond for relatively weak ablative material. In addition, it maintains an extremely low heat transfer, even for objects with large temperature differentials.

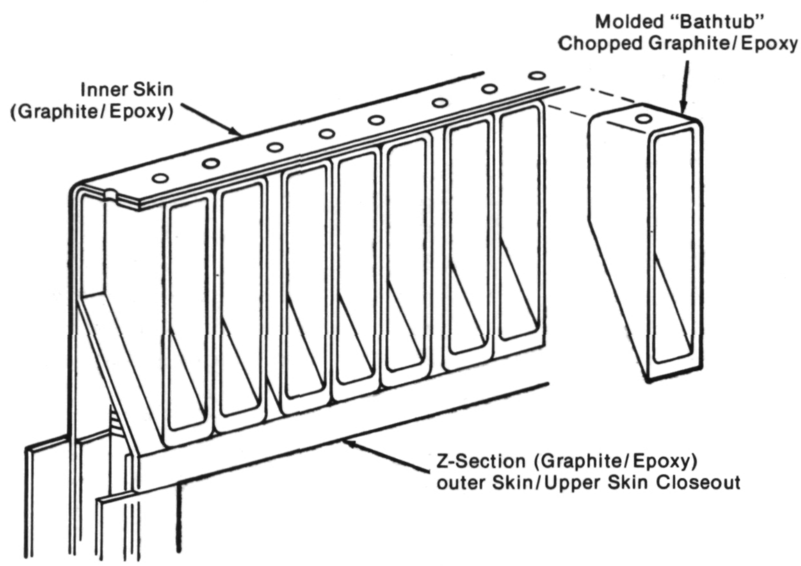
The plug (see figure), is a double insert. It consists of a standard metal insert in a threaded fiberglass block. This block then provides a relatively large shearing surface to an ablative material, such as closed-pore foam and some types of composites.

This innovation could be of use to manufacturers of insulation, furniture, and other products using ablative materials.

Source: W. P. Coburn of
Rockwell International Corp.
under contract to
Johnson Space Center
(MSC-15828)

No further documentation is available.

ADVANCED-COMPOSITE CHOPPED-FIBER FITTINGS



Chopped-Fiber Fittings

PROPERTIES OF UNCUT GRAPHITE FIBER

Fiber Type	Filament Diameter Microns	Filament Diameter Standard Deviation	Average Density lb/cu in.	Weight per Unit Length of Tow lb/in.×10^{-6}	Ultimate Tensile Strength† ×10^3 psi	Modulus Elasticity† ×10^6 psi	Specific UTS ×10^6 in.	Specific Modulus ×10^6 in.
A	7.9	0.42	0.0628	49	275-325	28-35	4.4-5.2	440-550
HM	7.5	0.30	0.0700	41	250-325	50-60	3.5-4.6	700-850
HT	7.8	0.35	0.0635	44	350-450	35-42	5.5-7.0	550-660

†Lower values are proven minimum values.

Chopped boron or graphite incorporated in an epoxy matrix can be made into advanced-composite joints that are light but strong, an advantage over heavier metal fittings.

The chopped fiber, in lengths of 6 mm (0.25 in.) and longer, is combined with an epoxy molding compound and pressed into "bathtub" molds. By varying the amount, length, or orientation of the chopped fibers, the mechanical properties of the composite are altered.

These fittings are used to attach a composite structure to an adjoining structure, in such a way that loads will be transferred from one structure to the other. In the figure, several of the fittings are used to join two sections of graphite/epoxy fiber. Similar arrangements could be useful in building and in the manufacture of ships and automobiles.

A table of the properties of uncut graphite fiber is given. The table includes a warning about airborne graphite fibers that could affect electrical equipment, since they are conductive.

Source: K. C. Dullea of
Rockwell International Corp.
under contract to
Marshall Space Flight Center
(MFS-24303)

Circle 11 on Reader Service Card.

CUTOFF-AND-MILLING MACHINE FOR HONEYCOMB CORE

A universal cutoff-and-milling machine has been developed for machining honeycomb cores and sizing beryllium sheet. The machine is relatively inexpensive to manufacture, and it will hold close tolerances over an entire working area.

The machine was conceived originally for cutting beryllium sheet. Previously, the sheet had to be sandwiched between heavy aluminum sheets before it could be bandsawed. This production tool next was used to machine sections of phenolic honeycomb core, by taping the core to the machine table with double-sided tape.

Other applications include: (a) sectioning Rene 41 panels, (b) machining plastics, (c) light milling of aluminum, (d) tape layup while fabricating composite panels, and (e) making accurate flat-pattern layouts.

Some of the outstanding features of the machine are:

a. Machining and cutoff may be done over a larger area while maintaining close tolerances.
b. The cutting head can change from a vertical to a horizontal cut simply by removing a pin.
c. Feed rates are continuously variable: up to 60 in./min horizontally and 10 in./min vertically (25 and 150 cm/min).
d. It can be converted quickly from machining to cutting and vice versa.
e. An automatic-feed indexing system can be added. The system synchronizes the feed index on the vertical axis with the position of the cutting head on the horizontal axis.

Source: W. P. Coppfer and H. B. Cain of
Martin Marietta Corp.
under contract to
Marshall Space Flight Center
(MFS-22357)

Circle 12 on Reader Service Card.

HONEYCOMB-CORE STRUCTURES IN CURVED SHAPES WITHOUT SPLICING

A new method of producing honeycomb-core structures involves heat shocking the honeycomb core [at 530 K (500° F) for 20 to 25 seconds], before it is fully cured, and forming it over a male plastic tool using a polyvinyl bag. The core then is placed in a female tool, covered with glass fabric, bagged with vacuum blankets, and cured under full vacuum for two hours at 465 K (375° F).

In another one-stage preheating technique, an ultralight potting compound is cured without cell-wall or face-sheet distortion. A new compound was developed, which produced a strong bond when applied as thin as 0.1 mm (0.004 in.).

The industries dealing with lightweight structures will find these methods efficient and reproducible. No splicing is necessary to form spherical or conical shapes, and there is a possibility of making unique design configurations.

Source: J. A. Scholl of
Rohr Corp.
under contract to
NASA Pasadena Office
(NPO-11036)

Circle 13 on Reader Service Card.

FABRICATION OF CARBON FILM COMPOSITES FOR HIGH-STRENGTH STRUCTURES

Fiber-composite materials exhibit very desirable physical and mechanical properties, because fine fibers have qualities that are vastly superior to bulk material of the same composition. Similarly, thin films often exhibit properties that are substantially different from bulk material. In view of this, it has been proposed that thin films of selected materials might be used to improve the physical and mechanical properties of laminated composites.

Preliminary investigations have shown that the density, elastic modulus, and tensile strength of carbon films increase dramatically as film thickness is reduced from about 3.8 μm to 0.13 μm. For example, carbon films have been prepared with an elastic modulus of 345×10^9 to 896×10^9 N/m^2 (50×10^6 to 130×10^6 psi) and ultimate tensile strengths of 2.8×10^9 N/m^2 (4.1×10^5 psi). The coefficient of thermal expansion for the carbon films has been found to be close to zero. Moreover, it has been found that the high moduli and tensile strengths are retained in multiple-layer deposits, when individual carbon layers are separated by intervening films of titanium.

These observations indicate attractive potential applications of carbon-film structural composites for the construction of microwave filters or optical instruments which are subject to wide temperature variations. The qualities of high strength and low density also imply applications of the composites in aerospace structures or in architectural structures where a high strength-to-weight ratio is desired. The experimental procedure used for the deposition of ultrathin films of carbon and titanium in alternating layers is described in the following paragraphs.

A substrate (e.g., inert fluorocarbon polymer), approximately 10 by 15 cm (4 by 5 in.) in area and 0.013 or 0.007 mm (0.003 or 0.0018 in.) thick, was cleaned ultrasonically in a detergent bath, followed by rinses in acetone, isopropyl alcohol, and deionized water. Then the substrate was supported in a metal frame which kept it taut by the application of a nominal longitudinal tension of 22 N (5 lb). The assembly was placed in a vacuum-deposition chamber and maintained at 150° to 200° C (302 to 392° F) for about 1 hour at a pressure of 2.7×10^{-3} N/m^2 (0.4 psi). The substrate was rotated at 60 rpm during deposition so as to form essentially equal thicknesses of film on its two sides. Electron-beam heating was used to evaporate source material; typically, the beam current was held at 150 to 200 mA. Deposition rate and final thickness were controlled and monitored by means of a quartz-crystal sensor and associated circuitry.

Composite films of carbon and titanium were formed by using a double-hearth system in the electron-beam deposition equipment. At preset intervals, the beam was switched for deposition of either carbon or titanium to produce the desired layered deposit.

The following documentation may be obtained at cost from:

National Technical Information Service
Springfield, Virginia 22151

Reference: NASA CR-1972 (N72-17537/LK), Physical Properties of Thin Films

Source: Peter R. Preiswerk and
Mike Lippman of
Astro Research Corp.
under contract to
Ames Research Center
(ARC-10613)

FELTED-PAPER BLEEDER MATERIAL FOR MOLDING COMPOSITE LAMINATES

A felted paper has been developed for use as an absorbent bleeder for collecting excess resin, air, and volatile matter from prepreg composite materials during cure of the materials. The paper improves the surface finish of the molded composite. It can be preformed to shape, which eliminates wrinkling while laminating complex parts.

The porous felted paper is separated from the top surface of the laminate by a porous fine-square-weave Teflon-coated fiberglass. The paper has a fine finish that is transferred to the laminate during cure, thus producing a relatively smooth bagged surface in comparison to laminates produced with fiberglass bleeders.

The paper material costs considerably less than either fiberglass or dacron conventional bleeder materials; and it is easier to handle than the fabric bleeders, because it does not unravel or fray, and it maintains its shape during layup. The bleeder material may be used in the fabrication of vacuum-bagged autoclave or hydroclave-molded reinforced-plastic components to provide major quality improvement and lower costs.

Source: J. S. Jones of
Rockwell International Corp.
under contract to
Marshall Space Flight Center
(MFS-24264)

Circle 14 on Reader Service Card.

BORON/EPOXY AND GRAPHITE/EPOXY NONDESTRUCTIVE TEST STANDARDS

The high-modulus and high strength-to-weight ratios of structures utilizing graphite or boron filaments incorporated into plastic matrices make them ideal replacements for many currently-used metallic structures. With the increased use of composites, their fabrication and testing will demand greater scrutiny through nondestructive evaluation than is presently provided for the less complex metallic structures.

A collection of nondestructive test standards have been prepared for Marshall Space Flight Center. They include detailed drawings of the standards that show general structures, types, and locations of simulated defects built into the panels. The panels were laminates with plies laid up in the 0°, ±45°, and 90° orientations. They contain either titanium substrates or interlayered titanium perforated shims.

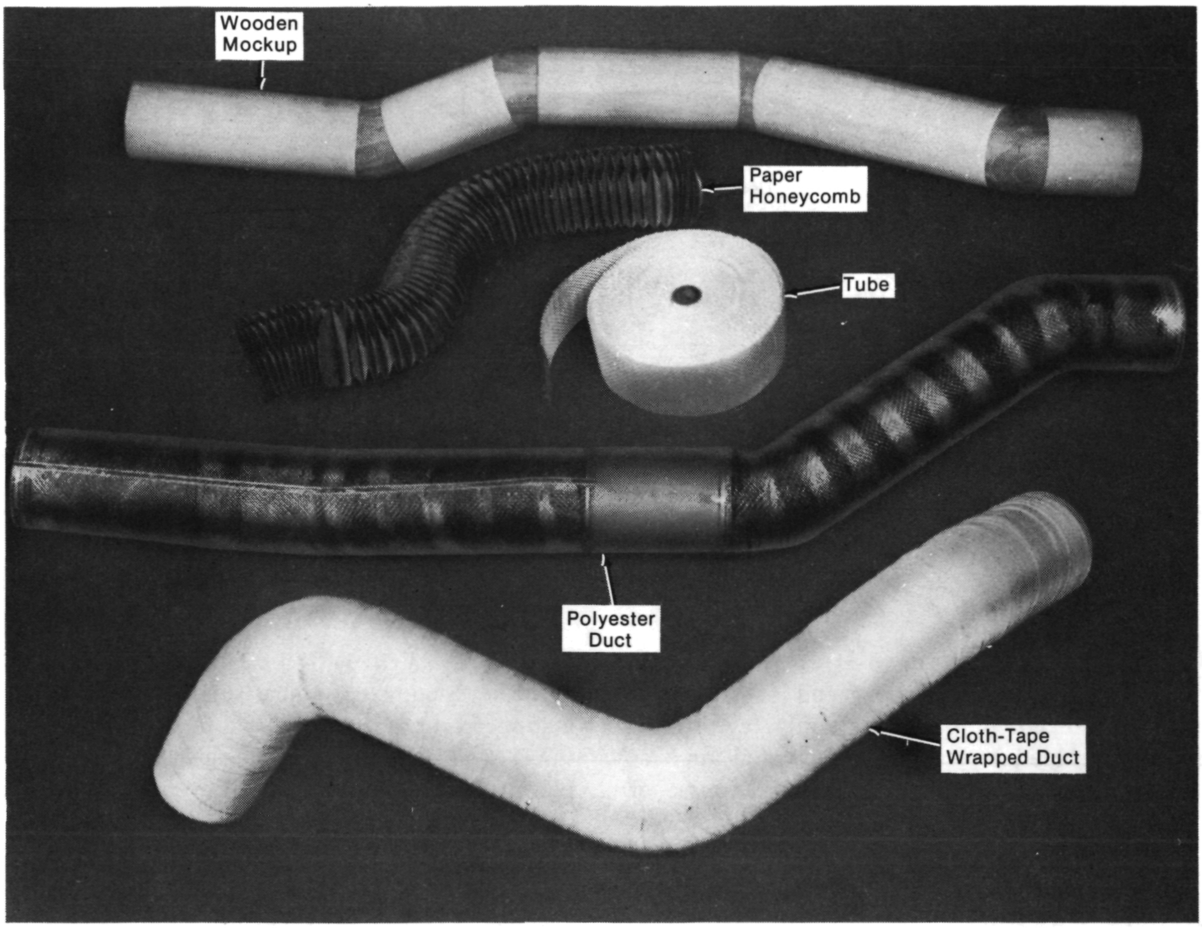
Panel thicknesses are incrementally increased from 2.36 mm (0.093 in.) to 12.7 mm (0.500 in.) for the graphite/epoxy standards, and from 2.36 to 6.35 mm (0.093 to 0.250 in.) for the boron/epoxy standards, except for the panels with interlayered shims which had a maximum thickness of 2.9 mm (0.113 in.). The panel internal conditions included defect-free regions; resin variations; density/porosity variations; cure variations; delaminations/disbonds at substrate bond lines or between layers; inclusions; and interlayered shims. Ultrasonic-pulse echo-C-scan and low-kilovoltage X-ray techniques are used to evaluate and verify the internal conditions of the panels.

Source: W. P. Pless and W. H. Lewis of
Lockheed-Georgia Co.
under contract to
Marshall Space Flight Center
(MFS-22055)

Circle 15 on Reader Service Card.

Section 3. Applications

TECHNIQUE FOR RIGIDIFYING PAPER-HONEYCOMB SIMULATED DUCTING



Comparison of Honeycomb-Paper-Core Duct with Polyester Duct

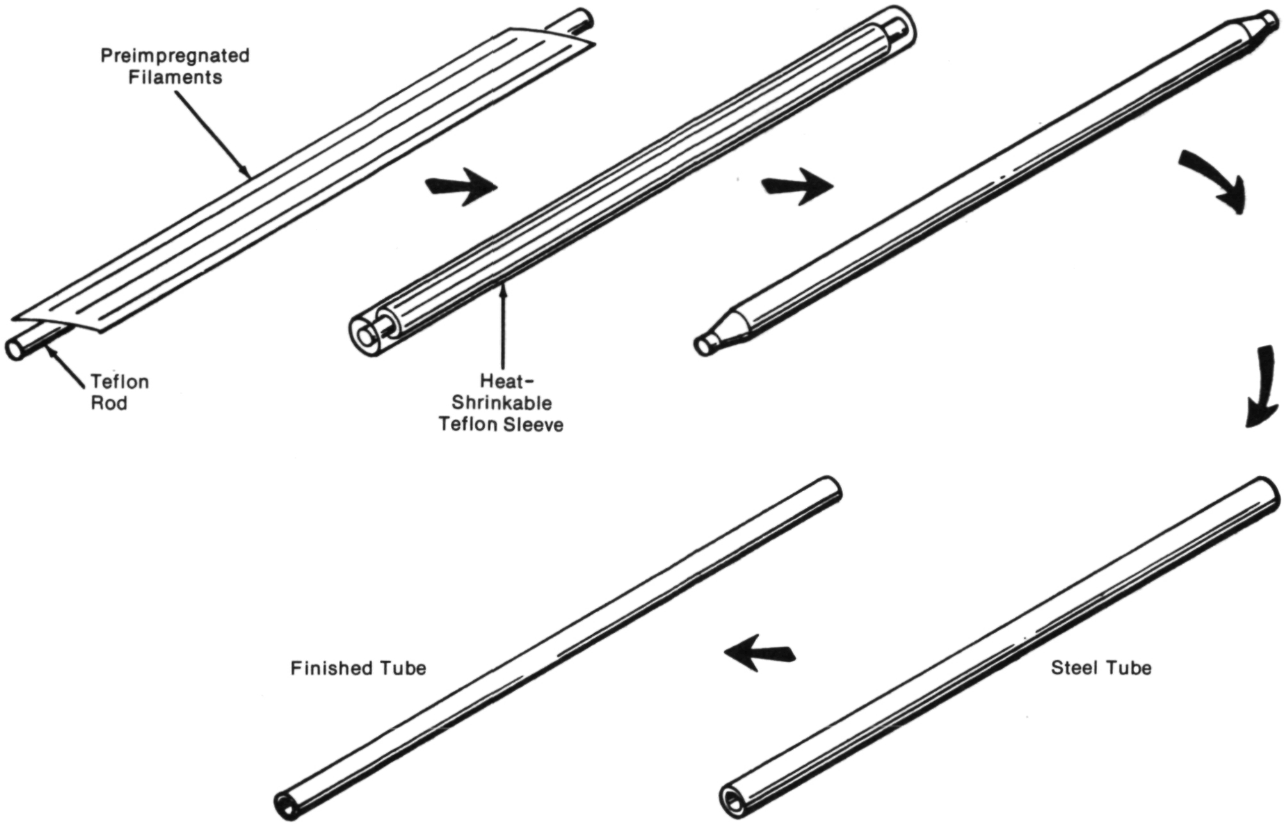
A low-cost method for rigidifying expanded honeycomb tubing and piping simulations has special advantages for irregular shapes.

In this paper/tape/resin system, a single spiral-wound glass-cloth-tape wrap is applied and coated with polyester. The photograph compares the cloth-tape wrap (in the foreground) with the more-rigid polyester-treated simulated duct shown just above it. A wooden mockup, a piece of expanded paper honeycomb, and a roll of the glass tape are shown at the top of the picture.

Source: P. R. McCune and R. A. Tucker of Rockwell International Corp. under contract to Johnson Space Center (MSC-19324)

No further documentation is available.

FABRICATION OF UNIAXIAL FILAMENT-REINFORCED EPOXY TUBES FOR STRUCTURAL APPLICATION



The high strength, high stiffness, and low density of filament-reinforced composite materials have stimulated considerable interest in their potential application to aerospace structures. During the course of research on composite materials, a unique process was developed for fabricating uniaxial filament-reinforced epoxy tubes of suitable quality for structural application.

The diagram illustrates the process. Strips of preimpregnated tape (a single ply of uniaxial filaments, rovings, or yarns embedded in a viscous epoxy resin) are cut and alined on a Teflon rod which serves as a removable mandrel. Care is taken to aline the filaments parallel to the longitudinal axis of the mandrel. The width of each strip is equal to the circumference of the tube, and plies are added until the desired wall thickness is obtained.

In the second step, a heat-shrinkable Teflon sleeve is slipped over the mandrel and tape. The diameter of the Teflon sleeve should be just large enough to permit the sleeve to be slipped over the tape without damaging the outer ply.

The third step consists of heating the Teflon sleeve with air from an electric heat gun. Complete shrinkage occurs at exposure to temperatures of 177° C (350° F), with partial shrinkage occurring at temperatures as low as 93° C (200° F). As the sleeve shrinks tightly on the composite material, air entrapped between the plies is squeezed out of the ends of the sleeve. In addition, the Teflon sleeve serves as a mold which forms a smooth outer surface on the filament-reinforced tube.

In step four, the entire assembly is inserted in a close-fitting steel tube, which prevents the mandrel

from sagging while the epoxy resin is cured at an elevated temperature. The steel tube and assembly are heated in a circulating-air oven for the final epoxy cure.

In step five, the assembly is removed from the steel tube, the Teflon sleeve is peeled from the outer surface of the tube, and the Teflon mandrel is extracted.

Tubes fabricated by this process have several advantages. They have very smooth inner and outer surfaces which are the result of molding against Teflon surfaces. The dimensional variation is less than the tolerances set for extruded aluminum tubing. The results of void content determinations indicate that composites are essentially void-free. Compressive and column buckling tests show that boron-epoxy tubes fabricated using the process described herein weigh approximately one-half as much as aluminum tubes designed for the same loadings.

Source: J. G. Davis, Jr.
Langley Research Center
(LAR-10203)

Circle 16 on Reader Service Card.

USE OF GRAPHITE FILAMENTS AS RESISTANT HEATERS TO CURE RESIN MATRICES

A technique for curing graphite filament composites is fast and adaptable to objects that are too large or cumbersome to be cured in an oven.

By means of copper conductor lines, an electrical current is passed through the graphite filaments that extend beyond the perimeter of the fixture to be cured. The graphite filaments act as resistance heating elements. The uniform heat will cure the material in less time than is usual in ovens. The resin must be precoated on the filaments, and the tooling be of some electrically nonconducting material, such as a plastic or asbestos board.

With this technique it is possible to heat composites up to 925 K (1,200° F) in less than one minute.

This technique is described in the following report:
"Heat Resistant Composite Structure Applications"
Reference: NASA CR-115713 (N72-28880/LK)

This report may be obtained at cost from:
National Technical Information Service
Springfield, Virginia 22151

Source: C. I. Yates of
Rockwell International Corp.
under contract to
Johnson Space Center
(MSC-14314)

SHEAR-RELIEF CONCEPT, FILAMENT-WOUND O_2 TANK

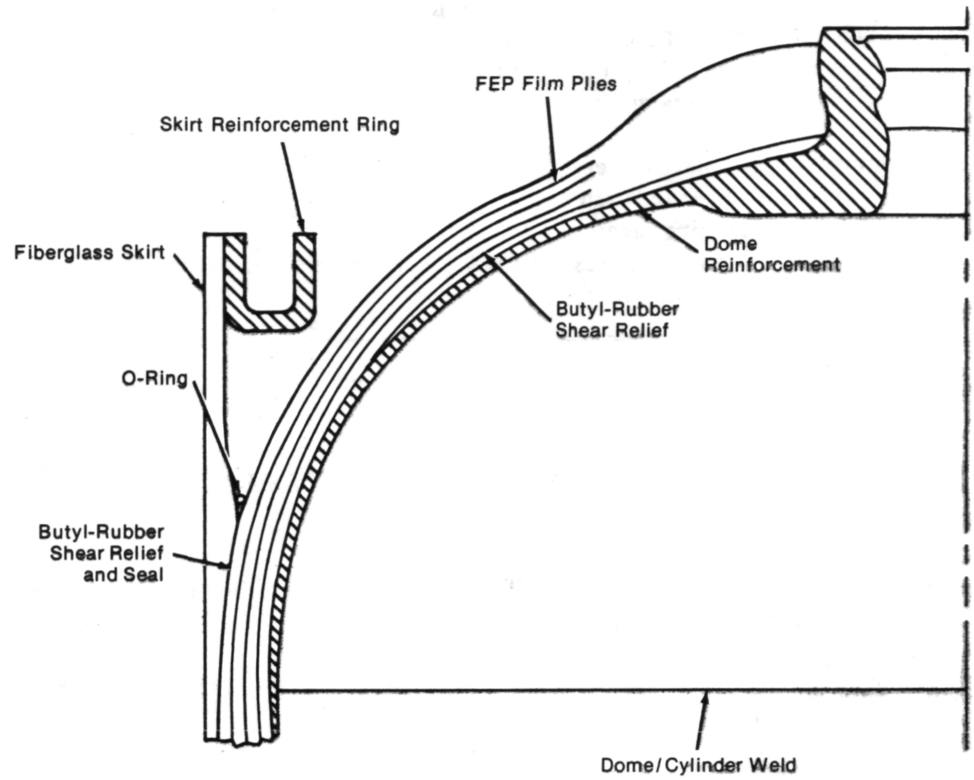


Figure 1. Oxygen Tank: Dome and Skirt After Filament Winding

A liner-overwrap combination, that can be optimized for each set of requirements, can be used for the storage of liquefied gas and the production of fuel cells, or any lightweight high-stress containers.

A plastic, reinforced with a glass fiber, is filament-wound over a stainless-steel cylinder, which ultimately serves as a liner or barrier for diffusion (see Figure 1). The new features here are a rubber strip and a plastic film inserted in the overwrap, in order to allow slippage rather than stress transfer. As a result, bending stress in the tank wall is reduced, while more of the fiberglass is put in tension, i.e., about 96 percent of the load of rated pressure.

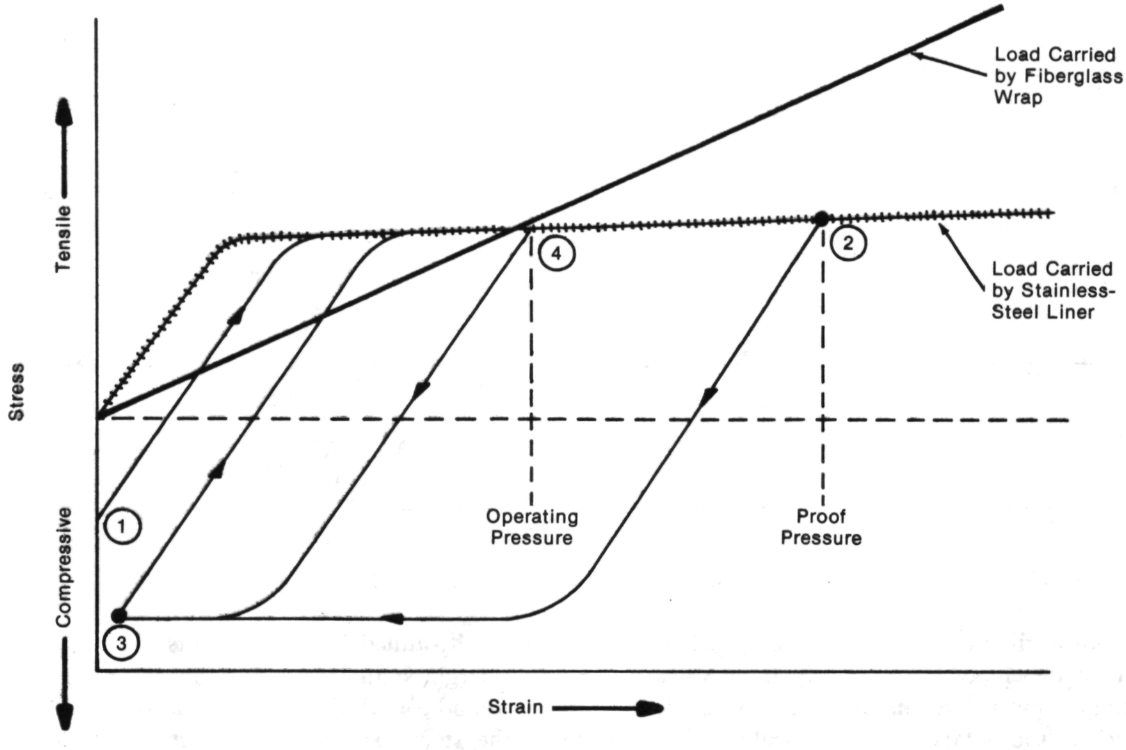


Figure 2. Stress/Strain Relationship for Stainless-Steel Liner

Point 1 is the compressive-yield point of the stainless-steel liner. The Line from 1 to 2 is the stress/strain relationship when the liner is subjected to proof pressure, and the Line from 2 to 3 is the stress/strain relationship when the proof pressure is reduced. Subsequent operating-pressure cycles follow the path between 3 and 4. Thus it can be seen how the overlap allows the compressive-yield point of the lines alone to be exceeded.

The stress/strain relationship for a 321 stainless-steel liner is shown in Figure 2. From this, it may be seen that the liner can strain beyond its tensile and yield points, during each cycle of pressurization.

Source: J. F. Schuessler and R. J. Dannenmueller of McDonnell Douglas Corp. under contract to Marshall Space Flight Center (MFS-22467)

Circle 17 on Reader Service Card.

BORON-FILAMENT REINFORCED-FIBERGLASS DRILL STEM

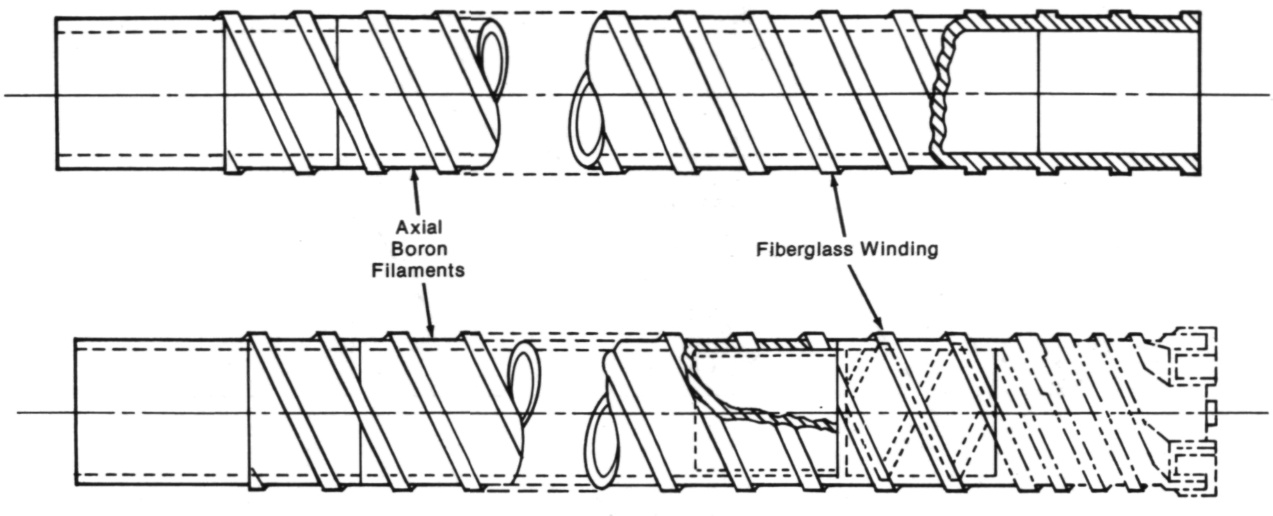


Figure 1. Boron-Filament-Reinforced Drill

A boron-filament reinforced-fiberglass drill stem has been developed for use on the lunar surface. This drilling tool is a novel application of boron-reinforced fiberglass and is favorably comparable to a titanium tool, and should be of interest to geophysical explorers in the petroleum and other industries.

The drill stem is shown in Figure 1. The tubular body of the drill stem has a sandwich construction, as shown in Figure 2. The innermost two layers are epoxy fiberglass; the glass filaments are helically wound at plus and minus 45° to the stem axis. Three layers of axially-alined boron filaments are wrapped over the fiberglass; and two additional layers of fiberglass are wound over the boron, again at plus and minus 45° to the stem axis. This configuration gives a nominal outer diameter of 2.51 cm (0.990 in.) with an inner diameter of 2.22 cm (0.875 in.).

The stem is fluted to transport drill cuttings to the surface. The fluter forms a 2.54-cm (1-in.) pitch double helix, wound around the stem body. The outer surface of the flutes are impregnated with silica powder to increase abrasion resistance. Each flute is

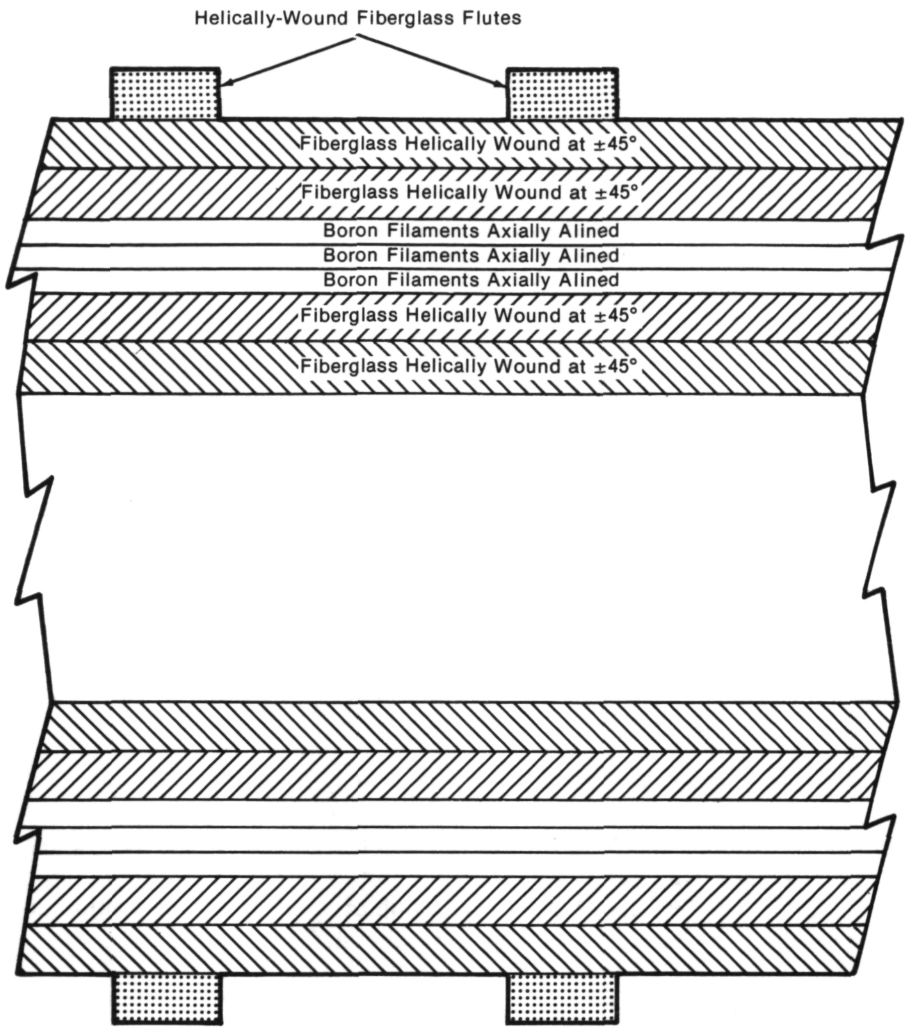


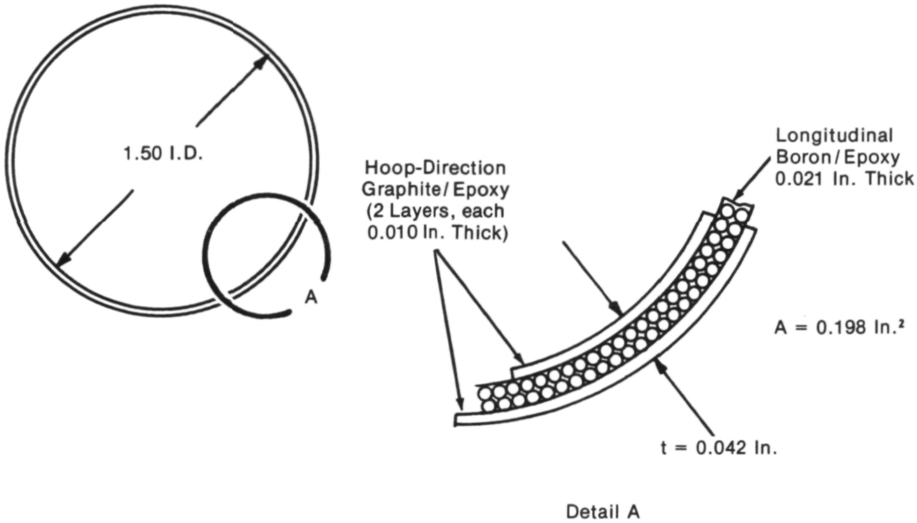
Figure 2. Cross Section of Drill Stem Showing Sandwich Construction

0.25 to 0.38 cm (0.10 to 0.15 in.) across and stands about 1.14 cm (0.45 in.) above the stem body, except in the area of the female joint (see Figure 1) where the stem body diameter is increased while maintaining the flute diameter constant. In the joint area, the flute depth is only about 0.025 cm (0.01 in.).

Source: M. G. Langseth, Jr., H. A. Gibbon, and R. S. Perry of Columbia University under contract to Johnson Space Center (MSC-13736)

Circle 18 on Reader Service Card.

LOW-CONDUCTIVITY COLUMN OF ADVANCED COMPOSITE EPOXY



Low-Conductivity Column of Advanced Composite Epoxy

A boron/graphite/epoxy composite column greatly reduces longitudinal heat conduction. This was shown by theoretical comparison with optimized equivalent columns made of various aluminum alloys and of stainless steel: The composite had less than 1 percent of the heat transfer of the optimum aluminum column and less than 2 percent of that of the stainless steel column.

The highly-orthotropic elastic modulus and the thermal conductivity of the fiber-reinforced composite were used to advantage. Proper selection of fiber orientation produced maximum stability and minimum heat transfer, so that the dual function of structural support and thermal isolation was accomplished. For small-diameter columns, graphite hoop-direction fibers were most practical (see figure).

Among the many possible uses for this column are: (a) in axial-load-carrying pipe supports (e.g., long-distance pipelines; (b) in heating, ventilating, air-conditioning, and aircraft equipment; and (c) in any support application where minimum heat transfer is required.

Source: E. C. Gadeen and
J. F. Darms, Jr.
Rockwell International Corp.
under contract to
Marshall Space Flight Center
(MFS-24317)

Circle 19 on Reader Service Card.

Patent Information

The following innovations, described in this Compilation, have been patented or are being considered for patent action as indicated below:

Graphite-Reinforced Aluminum Composite (Page 2) MFS-21077

Inquiries concerning rights for the commercial development of this invention should be addressed to:

Patent Counsel
Marshall Space Flight Center
Code CC01
Marshall Space Flight Center, Alabama 35812

Honeycomb-Core Structures in Curved Shapes Without Splicing
(Page 12) NPO-11306

This invention has been patented by NASA (U.S. Patent No. 3,658,974). Inquiries concerning nonexclusive or exclusive license for its development should be addressed to:

Patent Counsel
NASA Pasadena Office
4800 Oak Grove Drive
Pasadena, California 91103

Fabrication of Uniaxial Filament-Reinforced Epoxy Tubes for Structural Application
(Page 16) LAR-10203

This invention has been patented by NASA (U. S. Patent No. 3,607,495). Inquiries concerning nonexclusive or exclusive license for its commercial development should be addressed to:

Patent Counsel
Langley Research Center
Mail Stop 313
Hampton, Virginia 23665

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